NEAR EARTH ASTEROID RENDEZVOUS ORBIT PHASE TRAJECTORY DESIGN

J. K. Miller, E. Carranza, C. E. Helfrich, W. M. Owen and B. G. Williams

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Dr
Pasadena, CA 91109

D. W. Dunham, R. W. Farquhar, Y. Guo, and J. V. McAdams

Applied Physics Laboratory
Johns Hopkins University
Johns Hopkins Road
Laurel, MD 20723

Abstract

Trajectory design of the orbit phase of the Near Earth Asteroid Rendezvous (NEAR) mission involves procedures that depart significantly from those used for previous missions. On previous missions, the trajectory design involved finding a flight path that satisfied a rigid set of spacecraft and mission design constraints. A precise spacecraft ephemeris was designed well in advance of arrival at the target body. For NEAR, the uncertainty in the dynamic environment does not permit a precise spacecraft ephemeris to be defined in advance of arrival at the target body Eros. The principal cause of this uncertainty is limited knowledge of the gravity field and rotational state of Eros. As a result, the concept for NEAR trajectory design is to define a number of rules for satisfying spacecraft and mission constraints and to apply these rules to various assumptions for the model of Eros. This paper describes a trajectory design that applys to the physical model of Eros as currently understood. The final design will be completed after arrival at Eros and the gravity field and rotational state are determined.

Spacecraft and Mission Constraints

The spacecraft constraints that apply to Eros trajectory design include limits on fuel consumption, solar panel illumination and momentum wheel management. Other constraints define the flexibility and speed with which mission operations may be conducted. Probably the most important spacecraft constraint is to perform the prime mission within the allocated propellant budget. The propellant consumption constraint translates into about a 50 to 100 m/s delta velocity change during the orbit phase of the primary mission. This is a fairly generous allocation and should not be difficult to satisfy.

The most difficult spacecraft constraint to satisfy relates to solar panel illumination. Since the science instruments are fixed with respect to the spacecraft body, it is necessary to turn the spacecraft to point these instruments at Eros. In order to satisfy spacecraft power requirements, the solar panels cannot be turned more than about 30 degrees off the Sun-line. If the angle between the line to nadir and the plane perpendicular to the Sun-line is greater than 30 degrees, the nadir point cannot be imaged without turning the spacecraft more than 30 degrees. A coordinate frame is defined with the z axis pointing away from the Sun and the equatorial or x-y plane perpendicular to the Sun-line and is referred to as the Sun Plane-Of-Sky (POS) coordinate frame. In the Sun POS coordinate frame, orbits with inclination less than 30 degrees direct or greater than 150 deg retrograde will not violate the solar panel constraint permitting imaging of nadir from any point in the orbit.

Another important constraint relates to the time to conduct mission operations. In order to conduct the mission smoothly without resorting to round the clock operations, the minimum time between spacecraft propulsive maneuvers is limited to one week. The real limitation here is the time to compute accurate orbit determination solutions in support of propulsive maneuvers that are required to keep the spacecraft on course. The differential velocity change resulting from maneuver execution errors corrupts the orbit solution. A rapid redetermination of the orbit places a large amount of pressure on the Mission Operations team to deliver accurate data and process this data into reliable solutions for the spacecraft orbit. By allowing a minimum of one week between maneuvers, this pressure is considerably reduced. A further benefit is that the amount of data available for the orbit solution is increased and the data quality is increased. The more maneuvers that are performed, the more the orbit is corrupted and the more the quality of science is compromised. In addition, more risk to the mission is incurred because of poor trajectory control.

A trajectory design constraint related to orbit stability is that all low inclination orbits be retrograde with respect to the asteroid equator. Retrograde orbits are more stable because the faster relative motion of the spacecraft with respect to the asteroid tends to average out the effects of gravity harmonics. For this reason, synchronous direct orbits are particularly unstable since the spacecraft lingers over the same point on the asteroid's surface and may exchange enough energy to escape from or collide with the asteroid. In low orbit, even retrograde synchronous orbits may be unstable.

Science constraints on the trajectory design take the form of desires to obtain some particular orbital geometry and are generally not easily quantified. The requirement of the gamma ray spectrometer to obtain low altitude orbits will drive the trajectory design to achieve these orbits in a timely manner. The plan to stage the trajectory through a series of successively smaller circular orbits seems to satisfy most science and navigation requirements and makes the trajectory relatively simple to design. The general plan is to spend a specified amount of time in a series of circular orbits of pre-determined radius. This keeps the mission on schedule and enables a general imaging or mapping plan to apply for any Eros gravity field that may be encountered. Transfer orbits between the circular orbits may also provide a unique opportunity for science observations from a perspective different from the circular orbits. However, the need to get to desired circular orbits may also make the transfer orbits unattractive for science. In any event, the transfer orbits need to be designed to achieve circular orbits and only limited science constraints can be accommodated in these orbits.

Targeting Strategy

The general approach to the targeting strategy is to develop a broad set of objectives and compute a series of propulsive maneuvers that will steer the spacecraft in the direction of satisfying these objectives. This differs substantially from the traditional approach of defining a number of constraints and searching for the trajectory that globally maximizes some performance index. The NEAR approach is to compute a maneuver that satisfies a local set of constraints and then propagate the trajectory into the future. At the appropriate time, a minimum of one week in the future, the constraints are reevaluated and another maneuver is computed. This strategy is repeated until all the science objectives are achieved.

The spacecraft is first placed in an orbit that is in the Sun POS. The solar panel illumination constraint dictates that the spacecraft remain close to this plane for most of the mission. Otherwise, the solar panels would have to be turned too far off the Sun-line in order to image nadir. Also, staying in this plane for the first few weeks of the mission will minimize the effect of solar pressure on the trajectory and on the attitude control momentum management. This is particularly important at high altitudes where the solar pressure acceleration is a large contributor to the total acceleration and is much easier to model when the spacecraft is pointed directly at the Sun.

At the time of arrival at Eros, the spin axis of Eros is pointed away from the Sun. Since the Sun POS coordinate frame z axis also points away from the Sun, a retrograde orbit in the Sun POS will also be retrograde in the asteroid equator. This is the direction that is established for the initial orbits. As Eros moves in its orbit about the Sun, the Eros spin axis first points away from the Sun, then perpendicular to the Sun-line and then toward the Sun. The spin axis, which remains essentially fixed in inertial space, appears to rotate in the Sun POS frame. During the time that the spin axis is pointed almost directly away from the Sun, a retrograde orbit in the asteroid equator will be close to being in the Sun POS and thus suitable for imaging Eros without turning far off the Sun-line. Retrograde equatorial orbits are generally very stable and thus the orbit altitude may be lowered to 35 km for gamma ray spectrometer observations. As the Eros spin axis aligns perpendicular to the Sun-line, the Sun POS orbit results in a polar orbit with respect to Eros. This is also a stable orbit. A problem occurs when the Eros spin axis is about 45 degrees off the Sun-line. For these orbits, the node of the orbit plane with respect to the asteroid equator precesses at a fast rate sometimes approaching 5 degrees per day. During this transition zone, the spacecraft orbit must be actively controlled with maneuvers to keep the spacecraft within 30 degrees inclination of the Sun POS. As Eros spin axis rotates from perpendicular to the Sun POS to near alignment with the direction toward the

Sun, it passes through another transition region and then is placed in a retrograde equatorial orbit for the second time. This orbit is direct with respect to the Sun POS. Therefore, at the time the spacecraft is in the polar orbit and the Eros spin axis crosses the Sun POS, it is necessary to execute a "plane flip" maneuver sequence to reverse the direction of the Sun POS orbit from retrograde to direct. When this sequence is completed, the Eros equator orbit remains polar.

Targeting Algorithm

The trajectory design is accomplished by transforming the targeting strategy into a specific step-by-step procedure referred to as the targeting algorithm. The general approach is first to translate spacecraft and science constraints to geometrical parameters that may be computed directly from the spacecraft trajectory about Eros. When necessary, propulsive maneuvers are targeted to these trajectory-related geometrical parameters and spacecraft and science constraints are implicitly satisfied. Therefore, the success of the targeting strategy depends on our ability to define geometrical parameters that relate directly to mission constraints.

The geometric parameters of interest that may be closely related to mission constraints are distances from Eros and angular positions of various celestial bodies with respect to Eros. A problem with these angles and distances is that they vary rapidly as the spacecraft orbits Eros and are thus difficult to target. For the targeting to be successful, a set of parameters that vary slowly with time need to be defined. A convenient set of parameters is the classical orbit elements. The classical elements describe the size, shape and orientation of a spacecraft orbit about a central body that may be represented as a point mass. As long as the spacecraft acceleration is dominated by the central gravity, the classical orbit elements do not vary significantly. This is true during a large part of the NEAR mission. In high orbits the solar pressure becomes a significant perturber relative to the central body gravity and in low orbits the gravity harmonics cause the classical orbit elements to osculate. However, we may use the osculating orbit elements as short term predictors of spacecraft motion and thus control the trajectory and satisfy mission constraints by targeting to these parameters.

The classical orbit parameters of interest for targeting the NEAR trajectory may be separated into several general categories. The first category describes the size and shape of the orbit which relates directly to energy and angular momentum. The radius of periapsis and radius of apoapsis may be used to control the size and shape of the orbit. The semi-latus rectum and eccentricity may also be used for this purpose. These parameters also implicitly control the period of the orbit. The second category describes the orientation of the orbit in inertial space. The longitude of the ascending node, argument of periapsis, and inclination orient the orbit in inertial space. These angles may be computed in either the Sun POS or asteroid equator coordinate frames. The solar panel illumination constraint may be satisfied by keeping the inclination in the Sun POS coordinate frame less than 30 degrees. The asteroid equator coordinate frame may be used to target polar or low inclination orbits. The final category of orbit parameters, obtained by solution of Kepler's equation, are the times that the spacecraft arrives at various points in the orbit. The true anomaly of the spacecraft, which is the angle measured from periapsis, is also included in this category. These points are candidate maneuver placements. The times of periapsis, apoapsis and crossings of the line of nodes or reference planes are of interest for maneuver placement. In addition to the classical elements, the cartesian components of position and velocity in various coordinate frames may be used as target parameters. Also, the cartesian components may be mixed with classical elements to define target parameter sets.

The first step of the targeting algorithm is to determine the time of the next propulsive maneuver. The spacecraft orbit is propagated into the future and the values of the target parameters are computed. If a constraint is violated or a point is reached in the mission where it is desirable to change the orbit size or orientation, a complete set of orbit elements is displayed which describes the local region of the trajectory. We assume that the orbit elements are osculating slow enough that the conic trajectory propagation is sufficiently accurate. The conic orbit propagation is generally good for several orbits. A suitable maneuver placement point is selected which is normally at periapsis, apoapsis or the crossing of some reference plane. A precision trajectory is propagated to the nominal maneuver time obtained by solution of Kepler's equation. The osculating orbit elements are reevaluated and a few iterations may be required to determine the precise time of maneuver placement.

The next step is to select three target parameters that describe the post maneuver orbit. These parameters must be independent and include the maneuver point on the post maneuver orbit. The independence of the parameters may be verified by determining that the Jacoby matrix is non singular. For example the parameters peri apsis radius, apoapsis radius and eccentricity would not be independent because eccentricity may be determined from the other two. However, the parameters periapsis radius, apoapsis radius and inclination are independent and would be suitable for targeting. The inclusion of the maneuver point on the post maneuver orbit is a subtle condition to satisfy. An example would be transfer to a 35 km circular orbit from a maneuver placement at 50 km. Clearly this would not be possible because the 35 km orbit would not contain any point at 50 km. A more subtle example would be transfer to zero inclination. For this target parameter to converge, the maneuver placement point would have to be in the reference plane.

The third step is to determine the time to evaluate the target parameter constraints. Most of the time this is immediately after the maneuver. However for some orbits that are osculating severely, the constraints may be evaluated sometime in the future where it is desired to control some parameter of particular interest. For example, if we are trying to control the precise periapsis radius, the constraint may be evaluated at the nominal time of periapsis passage several orbits after the maneuver thus forcing the minimum radius to occur at this time.

The final step is to determine the finite burn velocity correction that satisfies the constraint parameters. The matrix of partial derivatives that relate the target parameters to the maneuver velocity components is computed by finite difference from precision trajectory propagations. This 3 by 3 matrix is inverted and multiplied times the required target parameter correction to obtain the delta velocity correction. The Newton-Raphson iteration is repeated several times until the desired target parameters are achieved.

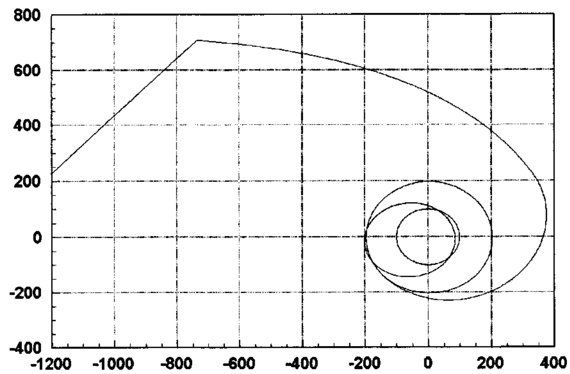
NEAR Trajectory Design

The design of the NEAR orbit phase trajectory involves repeated application of the above targeting algorithm. The model of Eros used for targeting is described in Reference 1 and the parameters are updated in Reference 2. A complete description of the NEAR mission is contained in References 3 through 7.

The orbit phase begins after a series of rendezvous burns that slow the spacecraft down from an approach speed of about 1 km/s to 5 m/s. The maneuver schedule resulting from the trajectory design is given in Table 1. The NEAR orbit phase trajectory is divided into 27 segments beginning with the Eros Orbit Insertion (EOI) maneuver (segment 0). Each segment begins with an OCM and the final segment ends on February 6, 2000, the nominal end of the mission.

Approach through 100 km Orbit

An orbit insertion maneuver is performed on January 10, 1999 at closest approach to transfer the spacecraft from an approach hyperbola with a periapsis radius of 1000 km to a highly eccentric ellipse with a periapsis radius of 400 km. When the spacecraft arrives at 400 km radius, a maneuver is executed to transfer the spacecraft to an orbit with a periapsis radius of 200 km. At 200 km radius, the orbit is circularized and science observations are carried out for about nine days. Two more maneuvers are executed to lower the spacecraft orbit to 100 km. The projection of the spacecraft orbit into the Sun POS coordinate frame is shown on Figure 1 for the initial orbits through 100 km radius. The view is from behind Eros looking toward the Sun. The spacecraft moves in a retrograde (clockwise) direction while the Eros rotation is counter clockwise when viewed from



Sun POS - km

Fig. 1 Approach Through 100 km Orbit

this direction. The initial orbit following the orbit insertion burn is in the Sun POS with a radius of periapsis of 400 km. The approach trajectory is targeted to cross the Sun POS at a radius of 1000 km periapsis on the side of Eros that results in a retrograde orbit. The orbit insertion burn is placed at the point where the approach hyperbola pierces the Sun POS; that is, the plane perpendicular to the Sun-line that passes through the center of Eros. The target parameters are radius of periapsis, true anomaly and inclination in the Sun POS.

Table 1. Orbit Phase Maneuver Segments

Date - Time (UTC)	Day	Seg.	Orbit (km x km)	Period (days)	Inc-ATE* (deg)	Length (days)	ΔV (m/s)
1/10/99 15:00	0	EOI	14,285 x 400	1788.3	135	10	8.45
1/20/99 15:59	10	1	401 x 200	14.8	137	7	0.74
1/27/99 16:40	17	2	205 x 195	8.0	138	6	0.29
2/5/99 16:40	23	3	205 x 85	5.0	138	10	0.51
2/12/99 21:48	33	4	102 x 99	2.9	146	10	0.94
2/22/99 23:59	43	5	510 x 98	15.1	62	5	4.05
2/27/99 21:48	48	6	500 x 100	14.8	157	10	1.08
3/9/99 19:02	58	7	165 x 50	3.2	154	6	1.77
3/15/99 18:19	64	8	56 x 49	1.1	158	15	1.41
3/30/99 08:35	79	9	55 x 33	0.8	169	26	0.98
4/25/99 04:34	105	10	35 x 35	0.6	178	46	1.43
6/10/99 06:24	151	11	55 x 34	0.9	177	7	0.43
6/17/99 15:09	158	12	51 x 49	1.0	178	8	0.32
6/25/99 08:27	166	13	50 x 49	1.0	147	7	2.17
7/2/99 18:06	173	14	52 x 49	1.0	120	20	3.74
7/22/99 04:10	193	15	50 x 40	0.9	123	7	1.65
7/29/99 20:42	200	16	46 x 42	0.8	91	25	2.88
8/23/99 08:02	225	17	105 x 42	1.8	90	14	2.64
9/6/99 06:45	239	18	515 x 99	15.3	49	6	3.70
9/12/99 03:09	245	19	490 x 99	14.4	90	7	0.87
9/19/99 08:41	252	20	105 x 40	1.8	94	8	1.37
9/27/99 04:56	260	21	42 x 38	0.7	91	9	0.86
10/6/99 01:50	269	22	54 x 42	0.9	122	8	2.24
10/14/99 04:44	277	23	50 x 46	0.9	126	14	3.72
10/28/99 16:33	291	24	51 x 45	1.0	142	14	3.60
11/11/99 21:25	305	25	49 x 35	0.8	177	7	2.19
11/18/99 23:23	312	26	34 x 33	0.6	178	79	0.57
2/5/00 23:59	391		End of Mission		Total Deterministic		ΔV=54.85

*ATE - Asteroid True Equator

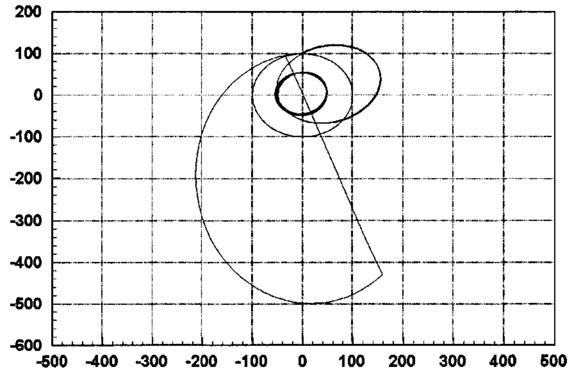
The target parameter constraints were computed on Jan 20, 1999 16:00:00. This targeting strategy provided a 10 day separation between maneuvers and resulted in the spacecraft arriving at a periapsis radius of 400 km on January 20 in an orbit with an inclination of 178 deg in the Sun POS. A 178 deg inclination orbit with respect to the Sun POS is 2 deg inclination retrograde or very nearly in the Sun POS. By targeting to orbit elements evaluated on January 20, 1999, the effect of solar pressure on the orbit elements was mitigated.

The first Orbit Correction Maneuver (OCM) is placed at periapsis of the initial transfer orbit. This maneuver is targeted to a periapsis radius of 200 km, apoapsis radius of 400 km and inclination of 177 deg in the Sun POS. OCM 2 is also performed at periapsis using the same targeting strategy and circularizes the orbit at 200 km radius.

After nine days, which is about one revolution in the 200 km orbit, a transfer orbit is computed to lower the spacecraft orbit to 100 km. At this time, the Sun POS inclination is about 171 deg. Recall that the Sun POS coordinate frame rotates as Eros orbits about the Sun and the orbit plane tends to remain fixed with respect to inertial space. In order to keep the spacecraft in the Sun POS, we must perform the orbit circularization maneuver at 100 km in the Sun POS. This may be accomplished by biasing the transfer orbit to a periapsis altitude of 84 km such that the spacecraft is at 100 km radius when the spacecraft crosses the Sun POS. This little trick saves a maneuver and the spacecraft orbit is circularized with another maneuver placed at the point where the spacecraft crosses the Sun POS at 100 km. In order to maintain one week separation between maneuvers, the spacecraft remains in the 200 km by 84 km transfer orbit for 2 and one half revolutions about Eros. The period of this transfer orbit is 5 days.

Subsolar Overfly through 50 km Orbit

An important science objective is to obtain infrared images of Eros at low phase angle. In order to obtain these images, it is necessary to place the spacecraft in an orbit that overflys the subsolar point. This may be accomplished by raising the inclination in the Sun POS coordinate frame to 90 deg. A convenient time to perform this over flight is early in the mission from the 100 km orbit. After the subsolar overfly, the spacecraft is parked in an elliptical transfer orbit for a week before the orbit is circularized at 50 km radius. Figure 2 shows the projection of the spacecraft trajectory in the Sun POS for the sub solar overfly through the initial 50 km circular orbit.



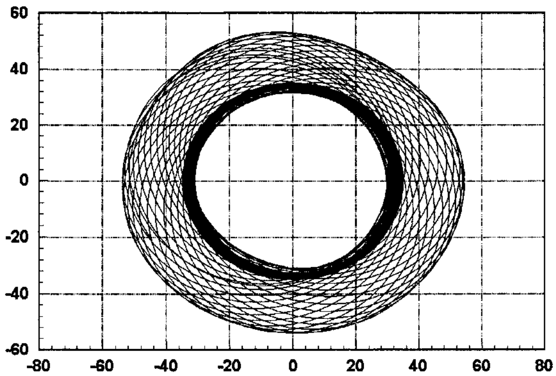
Sun POS - km

Fig. 2 Sub Solar Overfly Through 50 km Orbit

OCM 5 is targeted for overfly of the sub solar point. An orbit that flies over the subsolar point will also fly over the point opposite from the subsolar point or the anti-subsolar point. Therefore, if left in this overfly orbit, the spacecraft will fly through the shadow of Eros and violate an important spacecraft constraint to keep the solar panels illuminated. In order to avoid flying through the shadow and maintain one week separation between maneuvers, the spacecraft is placed in an orbit that flies over the subsolar point at a radius of about 150 km and continues out to an apoapsis altitude of about 500 km. The apoapsis radius is selected to give an orbit period of two weeks. After one week, at the point where the spacecraft crosses the Sun POS, OCM 6 is targeted for an elliptical return trajectory in the Sun POS and at a periapsis radius of 100 km. The periapsis radius of the return trajectory is selected to be 100 km in order to avoid being on an impact trajectory should the maneuver execution error associated with OCM 6 exceed the required accuracy. On return to periapsis, the spacecraft is parked in a 170 km by 50 km elliptical orbit for one week. This orbit is a compromise that enables the circular 50 km orbit to return to the Sun POS and avoids some of the rapid precession of the nodes associated with the 50 km orbit at this time. The spacecraft is then placed in a circular 50 km orbit where it remains until the Eros spin vector comes into favorable alignment for an equatorial orbit about Eros with a low inclination in the Sun POS. OCMs 6 through 9 are all targeted to radius of periapsis, radius of apoapsis and inclination in the Sun POS.

Transfer to Southern illuminated 35 km Orbit

A major science objective of the NEAR mission is to obtain low altitude gamma ray and x-ray spectrometer measurements of Eros. From a 35 km orbit, Eros fills the field of view and long integration times are required to obtain the data needed to characterize the composition of Eros. Figure 3 shows the projection of the spacecraft orbit on the Sun POS for the transfer orbit from 50 km to 35 km and the 35 km orbit.



Sun POS - km

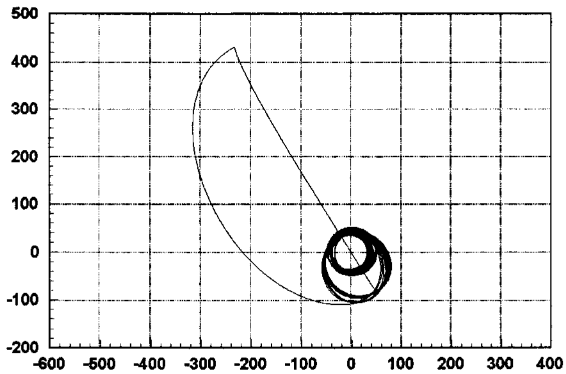
Fig. 3 Transfer to Southern Illuminated 35 km Orbit

OCM 9 is targeted for an elliptic transfer orbit from the 50 km circular orbit to a 35 km circular orbit. The actual dimensions of the nominal transfer orbit is 54 km by 32 km. For these low orbits, the conic elements are osculating such that the actual dimensions of the orbit vary in a complicated way. The dimensions of the orbit are contained by targeting the energy and angular momentum. The orbit elements periapsis and apoapsis radius are used to control energy and angular momentum. The orbital distances and velocity are controlled implicitly by managing energy and angular momentum. Even the energy and angular momentum are not conserved with respect to the two body motion of the spacecraft about Eros. The gravity harmonics act to exchange energy between the spacecraft orbit and Eros's rotation. In some cases, the gravity harmonics may act to eject the spacecraft from Eros orbit. The spacecraft actually receives a gravity assist from the gravity harmonics and the rotation of Eros. The basket weave appearance of the orbit shown in Figure 3 is caused by the rapid precession of the spacecraft orbit about Eros in inertial space and not by some artifact of a rotating coordinate system. After one month in the elliptic transfer orbit, the orbit is circularized at 35 km radius with OCM 10. For the transfer orbit and 35 km circular orbit shown on Figure 3, the target parameters are radius of periapsis, radius of apoapsis and inclination in the Eros equatorial coordinate frame.

Active POS Control, Polar Orbits and Plane Flip

As Eros moves in its orbit about the Sun, the Eros equatorial plane rotates from alignment with the Sun POS to being perpendicular to the Sun POS and back to alignment. This rotation of the planes with respect to one another occurs because the Eros spin axis remains fixed in inertial space while the Sun POS coordinate frame slowly rotates with Eros in its orbit around the Sun. During the time of the 35 km equatorial orbits, these planes are nearly aligned. As they change to an angle greater than about 30 deg, it is no longer possible to stay in an Eros equatorial orbit and at the same time satisfy the Sun constraint defined in the Sun POS. During the time that the angle between these planes is about 30 to 60 deg, the precession of the orbit about the Eros equator causes the Sun constraint to be violated if left unattended. Active POS control must be executed to prevent the Sun constraint from being violated. Eventually the angle between the planes approaches 90 deg and the spacecraft orbit may be transferred to a polar orbit with respect to Eros and at the same time satisfy the Sun constraint. In the polar orbit, the mission is once again interrupted for another overfly of the subsolar point. During this overfly sequence of maneuvers, the direction of the orbit in the Sun POS is reversed from retrograde to direct. This sequence of maneuvers is referred to as the plane flip. The projection of the spacecraft orbit on the Sun POS for the period of active POS control through the polar orbits and plane flip maneuver are shown on Figure 4.

OCM 11 transfers the spacecraft from the 35 km circular orbit to a 55 km by 35 km elliptical transfer orbit. The apoapsis radius of 55 km is set in anticipation of circularizing the orbit at 50 km and returning to the Sun POS. OCM 12 circularizes the orbit a week later at 50 km and the plane of the orbit remains in the Eros equatorial plane. A week later, the orbit plane is transferred to the Sun POS with OCM 13 and the period of active plane of sky control begins. Precession of the line of nodes with respect to Eros equator causes a break in the longitude of ascending node and a gradual increase in inclination as observed in the Sun POS. One week later or at the time the inclination reaches 30 deg, a maneuver is performed to flip the line of nodes in the Sun POS by 180 degrees. The target parameters for this maneuver are periapsis radius, apoapsis radius and the z component of velocity in the Sun POS. Reversing the z component of velocity is a device to flip the line of nodes 180 degrees. This strategy will result in the inclination with respect to the Sun POS decreasing from 30 degrees to zero degrees and then increasing again to 30 degrees as the node precesses with respect to the Eros equatorial plane. This strategy is repeated several times through OCM 17 where the plane flip maneuver sequence is executed. The strategy for the plane flip sequence is the same as executed previously for the subsolar point overfly only the spacecraft returns in a direct orbit. After the plane flip sequence is completed, the spacecraft returns to a polar orbit about Eros.

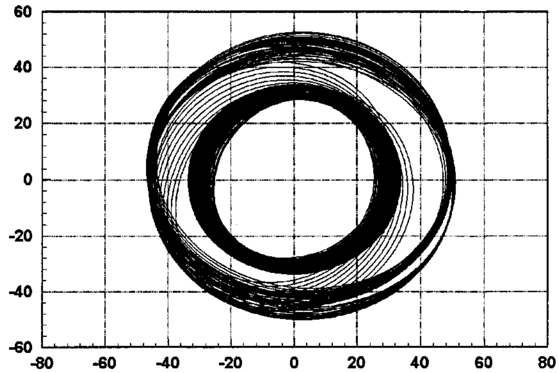


Sun POS - km

Fig. 4 Active POS Control, Polar Orbits and Plane Flip

Northern Illuminated 50 km and 35 km Orbits

Following the polar orbits, the spacecraft is placed in a 50 km by 45 km orbit for several weeks and active POS control is performed to keep the spacecraft from violating the Sun constraint. The targeting strategy is the same as used previously. When the north pole of Eros comes within 30 degrees of alignment with the Sun direction, a transfer to a 35 km equatorial orbit is executed. The spacecraft remains in the 35 km circular orbit until the end of the mission. The projection of the spacecraft orbit into the Sun POS for these orbits is illustrated in Figure 5.



Sun POS - km

Fig. 5 Northern Illuminated 50km and 35 km Orbits

Acknowledgment

The research described in this paper was carried out jointly by the Jet Propulsion Laboratory, California Institute of Technology and the Johns Hopkins University, Applied Physics Laboratory under contract with the National Aeronautics and Space Administration.

References

1. Miller, J. K., Weeks, C. J. and Wood, L. J., ''Orbit Determination Strategy and Accuracy for a Comet Rendezvous Mission,'' *J. Guidance, Control, and Dynamics*, Vol. 13, No. 5, Sep.-Oct. 1990., pp. 775-784.

2. Miller, J. K., Williams, B. G., Bollman, W. E., Davis, R. P., Helfrich, C. E., Scheeres, D. J., Synnott, S. P., Wang, T. C. and Yeomans, D. K., ''Navigation Analysis for Eros Rendezvous and Orbital Phases,'' *J. Astron. Sciences*, Vol. 43, No. 4, Oct.-Dec. 1995, pp. 453-476.

3. Farquhar, R. W., Dunham, D. W. and McAdams, J. V., ''NEAR Mission Overview and Trajectory Design,'' *J. Astron. Sciences*, Vol. 43, No. 4, Oct.-Dec. 1995, pp. 353-371.

4. Scheeres, D. J., ''Analysis of Orbital Motion Around

433 Eros,'' *J. Astron. Sciences*, Vol. 43, No. 4, Oct.-Dec. 1995, pp. 427-452.

5. Dunham, D. W., McAdams, J. V., Mosher, L. E. and Helfrich, C. E., ''Maneuver Strategy for NEAR's Rendezvous with 433 Eros'', Paper IAF-97-A.4.01 presented at the 48th International Astronautical Federation Congress, Turin, Italy, Oct. 6-10, 1997.

6. Yeomans, D. K., et al., ``Estimating the Mass of Asteroid 253 Mathilde from Tracking Data During the NEAR Flyby,'' *Science*, 278, pp 2106-2109, Dec. 19, 1997.

7. McAdams, J. V., Dunham, D. W., Helfrich, C. E., Mosher, L. E. and Ray, J. C., ``Maneuver History of the NEAR Mission: Launch through Earth Swingby Phase,`` Paper ISTS 98-c-23, 21st International Symposium on Space Technology and Science, Sonic City, Omiya Japan, May 24-31, 1998.